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Hydrocarbon Fuelled Pulse Detonation Engine Analysis

Con Doolan

Aeronautical and Maritime Research Laboratory
Defence Science and Technology Organisation, South Australia, 5108 AUSTRALIA

Abstract

A multiple combustor pulse detonation engine concept is analysed using a fourteen species chemical equilibrium model. Results are coupled with an analytical two-dimensional supersonic inlet model to provide estimates of operational performance. The analysis provides preliminary engine sizing and operational information of hydrocarbon fuelled pulse detonation engines for future tactical missile systems and boost-phase propulsion for hypersonic air-breathing engines.

Introduction

Detonation waves are a highly efficient means of combustion. They consist of a closely coupled shock wave and reaction zone which consumes the fuel typically thousands of times faster than what occurs in a flame. The major advantage of harnessing detonation waves is the ability to design an extremely compact and efficient engine. There have been several aero-propulsion concepts based on the detonation wave (see Kailasanath [1]), one of which is the pulse detonation engine (or PDE).

In essence, a PDE consists of a tubular combustion chamber with a fixed thrust wall at one end with the other end left open to act as an exhaust. The tube is filled with a combustible mixture and a detonation is initiated at the thrust wall which provides thrust until the wave has passed out of the tube and expansion waves have reached the thrust wall. Figure 1 illustrates the operating process of PDE's which can be summarised in four stages. The first stage consists of filling the engine with a detonable mixture and exhausting any products left from the previous cycle. The second stage involves initiating the detonation at the closed end. The detonation wave then traverses the engine in the third stage, providing thrust on the closed end. The wave is exhausted in the fourth stage with thrust termination occurring when expansion waves from the open end arrive at the thrust wall.

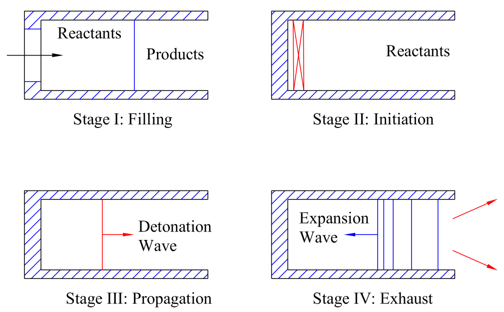

Figure 1. Operating process of an ideal PDE.

If the PDE concept is to be realised as an operational system, liquid hydrocarbon compounds will need to be used as a fuel. The benefits of liquid hydrocarbons include ease of storage and transport and high energy density. To achieve satisfactory energy density for a fuel such as hydrogen, complex and expensive cryogenic systems need to be developed, which impose a range of other problems for missile storage and the airframe. Hydrogen fuel, however, is able to produce a significantly higher specific impulse due to its small molecular weight.

Multiple Combustor Operation

As the PDE operates in an impulsive fashion, a simple way of maintaining a constant thrust on the vehicle is to use appropriately timed multiple combustors. Figure 2 illustrates the thrust timing sequence for a four-combustor PDE propulsion system. The thrust levels are offset to distinguish between each combustor. Considering thrust curve 1 in figure 2 shows the impulsive thrust instantly applied at zero time when the detonation is initiated. This is a simplification of the actual process of detonation initiation however, it is justified for the idealised study presented here. Thrust is maintained at the closed end for a time τ_T until the expansion waves from the open end reach the end-wall. At this point the next combustor is fired maintaining constant thrust on the vehicle. This process is continued for all the combustors in series. The recharge time τ_R for each combustor is therefore limited by the number of combustors and post-detonation wave fluid properties which in turn depends on the fuel type and supersonic inlet conditions.

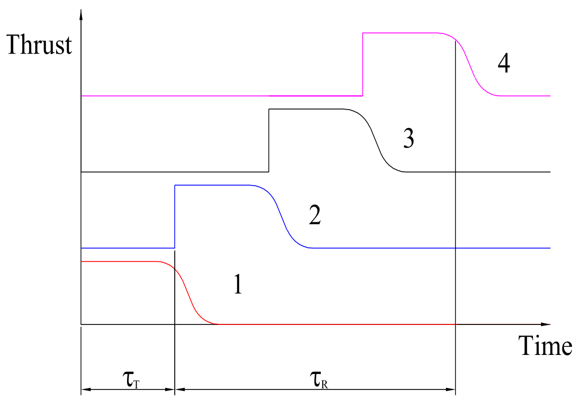

Figure 2. Firing sequence for a four-combustor PDE. The figure defines the thrust time τ_T and the recharge time τ_R.

Supersonic Inlet Model

A generic two-dimensional supersonic inlet was modelled to provide baseline conditions for PDE analysis. The inlet consists of a 10 degree isentropic compression corner followed by an oblique and normal shock wave to bring the air to subsonic conditions. The inlet was analysed using the gas-dynamic equations listed in Liepmann and Roshko [2]. Table 1 summarises the combustion chamber conditions at various altitudes using the U.S. standard atmosphere [3] and the supersonic inlet model.

M_∞	Sea Level		30 000 ft		65 000 ft	
	P (atm)	T (K)	P (atm)	T (K)	P (atm)	T (K)
1	1.00	288.15	0.30	228.71	0.06	216.05
2	4.94	458.16	1.47	363.65	0.27	343.52
3	18.30	734.78	5.43	583.21	1.02	550.93
4	44.65	1077.68	13.26	855.38	2.49	808.03
5	100.80	1642.46	29.93	1303.6	5.61	1231.4

Table 1. Combustion chamber conditions from supersonic inlet analysis.

Combustion chamber temperatures increase rapidly above Mach 4 at all altitudes and premixed hydrocarbon fuels will self ignite at temperatures above approximately 1100 K. Therefore the PDE is limited to a flight Mach number of about four. The analysis in this paper will use a baseline Mach 3 flight condition. This flight regime is very convenient for high speed tactical missiles. The PDE concept may also prove useful in providing the boost or acceleration phase for a scramjet powered vehicle. It should be noted that the PDE concept is not limited to a lower Mach number of unity. Operating concepts reviewed by Kailasanath [1] include a configuration where a self aspirating engine can operate from rest.

Detonation Model

A detonation wave is a shock wave coupled with an intense release of heat from burning fuel behind the wave. Detonation waves have been studied for many years and the theory behind them is very well known (see for example Kuo [4]). For a given initial mixture and thermodynamic condition, there exists a unique detonation condition known as the Chapman-Jouguet (CJ) point. This defines the detonation wave speed and the thermodynamic state of the products. To determine the CJ conditions using the combustion chamber conditions listed in table 1, a fourteen species chemical equilibrium reaction model is coupled with a gas dynamic detonation using the NASA Glenn CEA chemical equilibrium code [5]. The species used in the model include the fuel (either H_2, or C_8H_{18}) and N_2, O_2, H_20, CO_2, CO, OH, H, O, H_2 (as a product), NO_2, Ar, HO_2 and NO. The method for solving the chemical equilibrium state uses the minimization of Gibb's free energy with the Gordon and McBride polynomials [5].

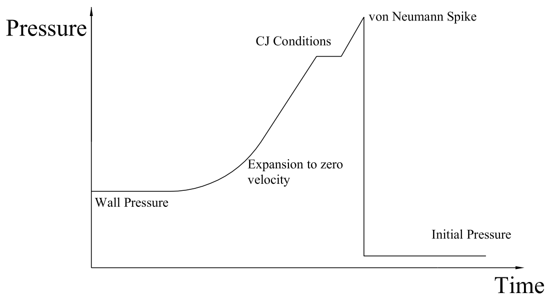

Figure 3. Variation of pressure behind an idealised detonation wave.

Figure 3 represents the pressure variation with distance behind an idealised detonation wave during the propulsive phase of a PDE cycle. The gas mixture is initially shocked to the high pressure von Neumann spike condition before relaxing to the CJ point. An expansion wave from the zero-velocity boundary condition at the wall reduces the gas pressure to the wall pressure which provides the thrust for the engine. The high pressure region at the front of the wave can also be made to do work by incorporating an expansion nozzle at the combustor exit. The addition of a nozzle is analysed by Cambier and Tegner [6] but will not be included here as this will affect wave transit times.

Once the detonation wave is initiated and travelling in the combustor, the endwall pressure (P_{wall}) is determined by expanding the CJ thermodynamic conditions to the zero-velocity boundary condition at the thrust wall. This can be determined from the ratio of sound speed at the wall (a_{Wall}) to the sound speed behind the detonation wave (a_{CJ}),

$$\frac{P_{Wall}}{P_{CJ}} = \left[\frac{a_{Wall}}{a_{CJ}}\right]^{\frac{2\gamma_{CJ}}{\gamma_{CJ}-1}} , \quad (1)$$

where γ_{CJ} is the ratio of specific heats after the detonation wave has passed. The speed of sound at the wall can be determined from the Riemann invariant,

$$a_{Wall} = a_2 - \frac{(\gamma_{CJ}-1)}{2} u_2 , \quad (2)$$

where u_2 is the gas velocity behind the detonation wave. Once the detonation wave has passed out of the combustor, the head of the expansion waves will travel with a velocity equal to a_{Wall} into the combustor.

PDE Performance Estimates

The relationship between the operating frequency (f) and thrust and recharge times can be deduced from figure 2 and is written,

$$\frac{1}{f} = \tau_T + \tau_R . \quad (3)$$

The thrust time is limited by the passage times of the detonation wave (velocity V_D) and the expansion wave in the tube,

$$\tau_T = L\left(\frac{V_D + a_{Wall}}{V_D a_{Wall}}\right) . \quad (4)$$

In a multiple combustor PDE, recharge time is limited by the number of combustors (N) and the thrust time,

$$\tau_R = (N-1)\tau_T . \quad (5)$$

Combining equations (3-5) gives an expression for the combustor length (L) in terms of the detonation conditions, operating frequency and the number of combustors,

$$L = \frac{1}{fN}\left(\frac{V_D a_{Wall}}{V_D + a_{Wall}}\right) . \quad (6)$$

Specific impulse is the most common measure of performance for aero-propulsion systems. It is defined as the ratio of thrust to mass flow rate of fuel. It is normalised by the gravitational acceleration (g) when using SI units. In this study, the thrust is defined as the product of wall pressure and thrust wall area. The mass flow rate ($\dot{m}_f$) can be deduced from the ideal gas law and the mole fraction of fuel consumed in each propulsive detonation,

$$\dot{m}_f = fN\frac{X_f \bar{M} P_R A_c L}{RT_R} \qquad (7)$$

where X_f is the mole fraction fuel in the recharge mix, P_R is the pressure of the recharge mixture, A_c is the area of the thrust wall, R is the universal gas constant and T_R is the recharge mix temperature. Using equations (6) and (7) with the definition of thrust gives an expression for the specific impulse which is independent of the number of combustors and the operating frequency,

$$I_{sp} = \frac{P_{Wall}}{P_R}\left(\frac{V_D + a_{Wall}}{V_D a_{Wall}}\right)\frac{RT_R}{gX_f \bar{M}_f} \quad . \qquad (8)$$

By using the results of the supersonic inlet analysis with the chemical equilibrium model with the mole fraction of fuel set for stoichiometric conditions, the specific impulse for an airbreathing PDE system is plotted in figure 4. Note that this is for a perfectly timed system of multiple combustors operating at sea level. Results for hydrogen and octane fuels are presented which highlights the performance penalty for using a liquid hydrocarbon. The performance of the smaller chain hydrocarbons (e.g. propane, ethylene etc.) will lie between the two curves shown. Higher performance using liquid hydrocarbons may also be achieved by using speciality jet fuels such as JP-10. Also shown in figure 4 are typical values of specific impulse for a solid propellant rocket and a hydrocarbon fuelled ramjet. The results show that the PDE promises to provide significant increases in performance over these systems.

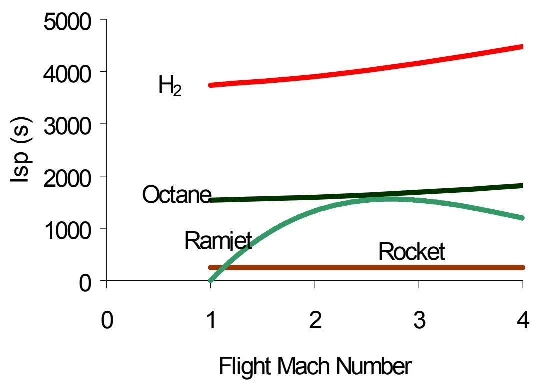

Figure 4. Specific impulse for perfectly timed, multiple combustor airbreathing PDE's operating with hydrogen and octane fuels at sea level. Typical values for a solid propellant fuelled rocket and a hydrocarbon fuelled ramjet are shown for comparison.

The size of the required engine system can be estimated from the above arguments in conjunction with an estimate of the vehicle drag,

$$\frac{L}{D} = \left\{\frac{V_D a_{Wall}}{V_D + a_{Wall}}\right\}\left(\frac{2\rho_\infty C_D A_m}{\pi P_{Wall}}\right)^{-0.5}\frac{1}{fNa_\infty M_\infty} \quad , \qquad (9)$$

where D is the combustion chamber diameter, ρ_∞ is the freestream density, C_D is the drag coefficient of the vehicle, A_m is the frontal area of the vehicle or missile and a_∞ is the freestream speed of sound.

Figure 5 plots the length to diameter ratio for some multiple combustor PDE's against operating frequency. In this plot, it is assumed that the PDE's are fitted to a missile cruising at sea level with a flight Mach number of three. The diameter of the missile is assumed to be 1 m. The drag coefficient was estimated from the missile aerodynamic tables in Sutton [7].

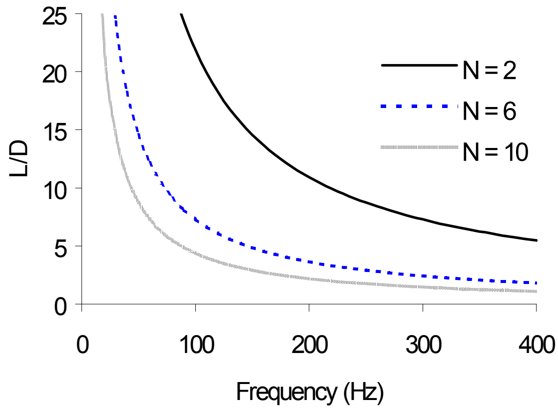

Figure 5. Length-to-diameter ratio for multiple combustor airbreathing PDE's versus operating frequency. It assumed that the PDE's are fitted to a 1 m diameter Mach 3 missile cruising at sea level.

A highly compact engine requires a small length-to-diameter ratio. From figure 5, it can be seen that operating frequency has a large dependence on engine size which approaches a limit as the operating frequency is increased to infinity. A practical maximum length-to-diameter ratio is 6.5, hence for a six combustor PDE the minimum operating frequency is 110 Hz.

The time to recharge can also be calculated for multiple combustor PDE's from a simple relation deduced from equation 3 and figure 2,

$$\tau_R = \frac{1}{f}\left(\frac{N-1}{N}\right) \quad . \qquad (10)$$

Equation 10 is plotted in figure 6 which also includes an estimated recharge time for a Mach three missile cruising at sea level. This is determined from the supersonic inlet analysis and is based on the gas flow available after the terminating normal shock at the inlet throat. The intersection between this line and the recharge time curve yields the maximum operating frequency for a given flight condition and number of combustors. For the example used here, a six combustor PDE system has a maximum operating frequency of 260 Hz.

Therefore from figures 5 and 6, given a size constraint of the PDE, the operating frequency range can be estimated. For L/D = 6.5, this range is 110 Hz to 260 Hz. The operating frequency range becomes more restrictive the more compact the engine becomes. For example, an engine with L/D = 3.6 has an operating frequency range between 210 Hz and 260 Hz.

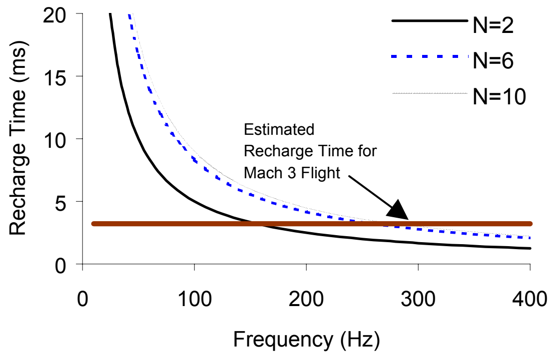

Figure 6. Recharge time for multiple combustor airbreathing PDE's versus operating frequency. Also shown is the estimated recharge time for a Mach 3 missile cruising at sea level.

Discussion

The above analysis shows that the pulse detonation engine can theoretically out-perform conventional rocket technology and hydrocarbon fuelled ramjets. There are however, many issues that need to be resolved before an operational system is feasible. A liquid hydrocarbon fuelled PDE has been operated by Brophy et al [8] but at a reduced rate of 5Hz. The challenge to engineers and scientists is to increase the operational frequency to in excess of 110 Hz .

One of the critical areas that needs to be resolved is detonation initiation. In order to achieve the required thrust a relatively large diameter PDE will be required. For the six combustor example analysed above operating at Mach 3, this equates to a chamber diameter of 150 mm. Initiation by direct means requires approximately 10^6 times more energy than what is required to initiate a flame by a spark [9]. A flame can transition to detonation in a combustion/deflagration wave coupling process known as deflagration-to-detonation transition (DDT), however this can take many tube diameters and hence increase engine size and reduce the maximum operating frequency. There have been a number of methods put forward to overcome this problem such as laser focussing [10] or hot jet penetration [11]. A technique which has produced promising experimental [8] and computational results [12] is the predetonator concept. Here, detonation is initiated in a smaller diameter tube by a low energy source and is allowed to transition to detonation. As the tube diameter is small, detonation can occur in a much reduced length. The predetonator is coupled to the main combustor and the smaller detonation wave diffracts into the combustion chamber initiating the mixture. Experimental [8] and computational [12] results have shown this technique can work if a more energetic fuel/oxygen mixture is used in the predetonator. This has implications for the overall engine specific impulse and further work is required to understand the diffraction process to adapt the predetonator to a purely airbreathing engine.

Other areas which need research attention for the PDE involve structural and heat transfer considerations. The repetitive shock loading of the detonation wave excites flexural stress waves in the combustor wall and if the operation frequency approaches this flexural wave frequency very high stresses will occur [13]. This has implications for the required wall thickness and weight of the combustor. Heat transfer from the PDE combustor remains to be quantified. The level of heat transfer from this system may be intense but could be utilised to "crack" the hydrocarbon fuel before detonation in the combustor. This would have the advantages of reducing the chemical induction time, increasing the ease of initiation and enhancing the fuel/air mixing and atomisation processes.

Conclusions and Future Work

This paper describes the PDE concept and provides an analysis technique which allows the determination of important operational parameters. A fourteen species thermochemical equilibrium model is coupled with a supersonic inlet analysis to provide details on specific impulse, engine size (L/D), operating frequency and recharge time. The model is based upon a specially timed multiple combustor system which provides constant thrust to the propelled vehicle.

Results show that the pulse detonation engine may provide at least an order of magnitude increase in specific impulse over conventional solid rocket motors. The analysis also suggests that operating frequency be above 110 Hz to achieve a practical, compact engine but will be limited to 260 Hz due to recharge times.

The flight envelope of a PDE powered vehicle ends at approximately Mach 4 due to auto-ignition of fuels at the high combustor temperatures expected. This flight range is convenient for the next generation of high performance missiles. The PDE may also be used as a boost motor for high Mach number airbreathers such as supersonic combustion ramjets.

The analysis presented here is useful for preliminary engine sizing but makes a number of assumptions which should be rectified for future operational analyses. Initiation of the reactants remains a significant technological challenge. The incorporation of a chemical kinetics model into the analysis will provide information on the effects of DDT and predetonators on engine performance. Also, the effects of viscosity and turbulence has not been included. Fluid viscosity will affect skin friction, heat transfer and detonation transition and should be included in the computational model at some stage. Multidimensional effects also need incorporation.

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